REPAIR OPTIONS FOR AIRFRAMES

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1. INTRODUCTION

Maintaining the airworthiness of in-service aircraft and at the same time keeping the maintenance cost low is of prime concern to the operators and regulatory authorities. In order to keep maintenance cost low, right decisions need to be made regarding replacing or repairing the in-service damaged components. The choice between replacing or repairing a structural component is governed by a number of factors such as the availability of spares, duration a structural component is expected to be in service, feasibility of repair, repair meeting structural integrity requirements, and inspection requirements for the repair. If it is economical to repair the component then the optimum repair design needs to be selected.

This paper discusses structural life enhancement techniques along with the state-of-practice methods of repairing metallic and composite structures. Applications of advanced repair methods such as composite patch repair of cracked metallic structures are discussed. Available computer codes for designing repairs are briefly described.

2. STRUCTURAL LIFE ENHANCEMENT OPTIONS

Stress levels, load spectrum, environment, structural details and the material of the structural component, govern the life of an aircraft structure. Under certain loading and environmental conditions a crack may initiate and propagate in a metallic structural component or environmental conditions may cause severe corrosion in the component. Depending on the structural details, the crack or corrosion damage may result in a catastrophic failure or costly repairs. A logical preventive method is to retard the initiation and growth of the cracks by pre-stressing so that the cracks do not result in catastrophic failure before the useful life of the structure. In certain cases this may not be feasible and a structure may have to be repaired to meet the useful life requirements. In addition, the in-service damage due to foreign objects in both metallic and composite structures frequently requires repairs so that the structure is able to carry the required load. Two commonly used techniques of structural life enhancement (Reference 1) by prestressing and repairs are summarized in Figure 1.

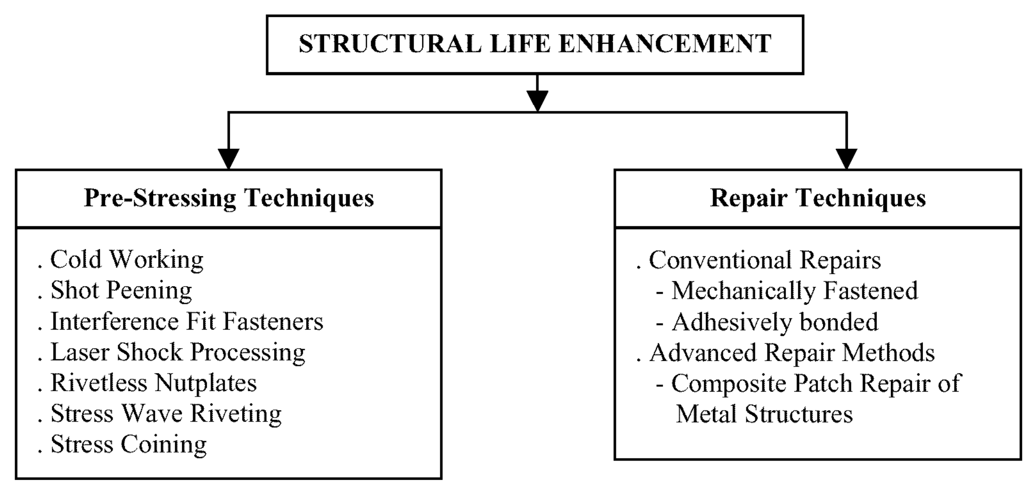

Figure 1. Life Enhancement Techniques

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Prestressing techniques to enhance structural life are generally used before a problem has occurred. In the design and analyses process, if a component or some parts/areas of a component are not able to meet design life requirements, prestressing process may be used for these locations to meet service life requirements. In case of in-service aircraft, if fleet data indicates cracking problems in certain areas, these areas may be subjected to prestressing process to enhance life before cracks initiate.

Life Enhancement Through Pre-stressing Techniques

In this technique a residual compressive stress field is created at highly stressed locations such as holes where cracks are likely to initiate. Subsequent inflight loads have to overcome the compressive stresses in order for the cracks to initiate and propagate. Some prestressing techniques have been fully developed while others are still in the development stage and have shown good promise to enhance structural life. The applications of these techniques to in-service aircraft are shown in Figure 2. The figure also shows the locations where these techniques are applied (e.g. whether the technique can be used at the manufacturing line, depot or field). The analysis methodology that can be used for life predictions is also shown in the figure. The level of verification testing required for successfully implementing the technique is also given in the figure. The extent of life enhancement achieved through these techniques is discussed in Reference 1.

PRE-STRESSING TECHNIQUE	**IN-SERVICE APPLICATIONS**	**LOCATION WHERE PERFORMED**	**ANALYSES METHODS**	**REQUIRED TESTING**
COLD WORKING	T-38, F-5, F-16, JSTARS F-18, F-111, C-141, 747	MANUFACTURING LINE, DEPOT AND FIELD	EQUIVALENT INITIAL FLAW(EIF), FATIGUE LIFE FACTOR(FLF)	MINIMUM
SHOT PEENING	T-38, F-5, F-18, F-14, 737,747,C-130,B-1	MANUFACTURING LINE, DEPOT AND FIELD	EIF, FLF	MINIMUM
INTERFERENCE FIT FASTENERS	T-38, F-5, F-18, 747	MANUFACTURING LINE, DEPOT AND FIELD	EIF, FLF	MEDIUM
LASER SHOCK PROCESSING	NONE KNOWN	MANUFACTURING LINE	DEVELOPMENT REQUIRED	SUBSTANT-IAL
RIVETLESS NUTPLATES	F-22, T-38	MANUFACTURING LINE, DEPOT AND FIELD	EIF, FLF	MEDIUM
STRESS WAVE RIVETING	F-14, A6E	MANUFACTURING LINE AND DEPOT	EMPIRICAL	MEDIUM
STRESS COINING	F-18, DC-8, DC-9, DC-10	MANUFACTURING LINE AND DEPOT	EMPIRICAL	MEDIUM

Figure 2. Prestressing Life Enhancement Techniques Applications

Life Enhancement Through Repairs

Structural life enhancement techniques through repairs for in-service fatigue, corrosion and foreign object damage (FOD) have been well established for metallic aircraft. With the increasing use of composites for improved structural efficiency, these methods have been developed for composite materials. However, there are basic differences between the damage types and their behavior in composite and metallic materials (Ref. 2-4). The basic differences between the behavior of metals and composites need to be understood so as to design proper repairs for metallic and composite structures. Figure 3 shows a comparison of typical metal and composite fatigue behavior under fighter aircraft wing spectrum loading. The data are plotted for each material's most sensitive fatigue loading mode, which is tension-dominated (lower wing skin) for metals and compression-dominated (upper wing skin) for composites. The figure shows that composite fatigue properties are far superior to those of metal.

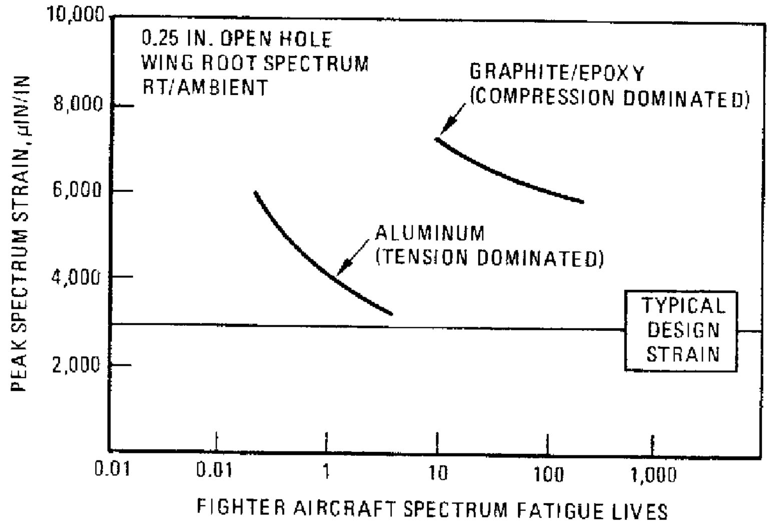


Figure 3. Comparison of Fatigue Behavior of Metallic and Composite Materials

A major consideration in the design of composite structures is the in-service impact damage. Impact damage occurs during ground handling, take-off and landing, and in-flight due to foreign objects. Hard objects (e.g. tool drops and runway debris) may cause impact damage and soft objects (e.g. bird impacts that occur at low altitude during take-off and landing). The impact damage caused by tool drops, etc. is termed as low velocity damage. Considerable reduction in compression strength may occur due to low velocity damage that is not visually detectable on the impacted or other external surfaces. The non-visual damage may cause internal damage in the form of delaminations between plies, matrix cracking, and fiber breakage. The longitudinal cross-section of an impact-damaged panel is shown in Figure 4. The damage due to impact is influenced by the factors such as laminate material properties, size of the laminate, support conditions, substructure, impactor size and shape, impactor velocity, impactor mass, impact location, and environment (Reference 5).

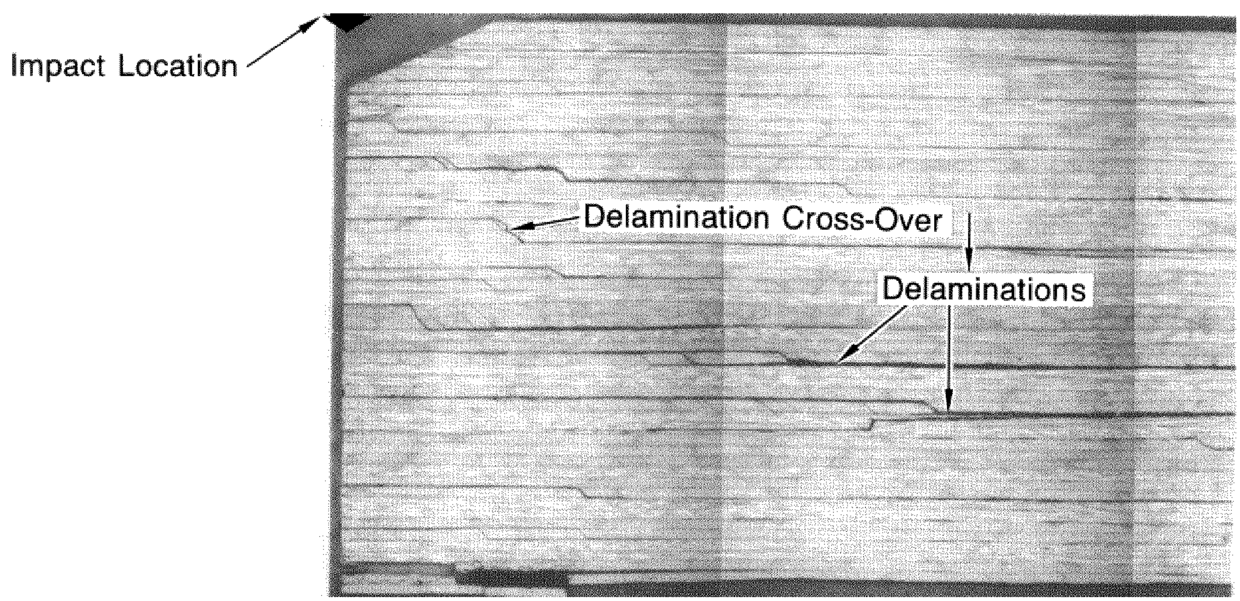


Figure 4. Impact Damage in Composites

Experimental data have shown (Figure 5) that impact damage can cause significant loss in strength. The degradation in compression strength is more severe than tension strength due to the delaminations between the plies caused by the impact damage (Reference 4).

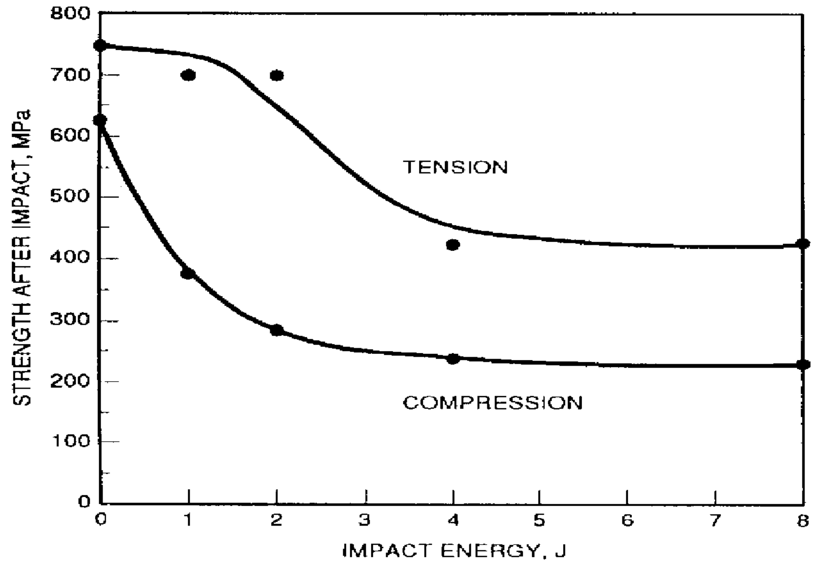


Figure 5. Strength Degradation Caused by Impact Damage

3. DAMAGE EVALUATION AND REPAIR CONCEPT SELECTION

The first step in designing any repairs is to evaluate the extent and nature of damage. Commonly occurring in-service damages in metallic and composite structures are shown in Figure 6. The overall process involved in damage evaluation and making repair decisions for a metallic and composite structure is outlined in Figure 7. Once the nature and extent of damage is found it is important to determine the effect of damage on structural integrity. If in a metallic structure, the damage found is a small crack that is much smaller than critical crack length, the repair may be performed by enlarging the hole to remove the crack and using an oversize fastener. In such cases, a revised damage tolerance analysis needs to be performed and new inspection requirements imposed for that location.

Metallic Structures	Composite Structures
Fatigue Cracks	Delaminations
Corrosion	Impact Damage
Stress Corrosion	Foreign Object Damage
Foreign Object Damage	

Figure 6. In-service Damage Types in Metallic and Composite Structures

The type of repair to be performed will be determined by the following factors-

1. Type of structural material to be repaired (metal, composite, sandwich construction)
2. Type of structural component to be repaired (skin, spar, rib, longeron, etc.)
3. Type and extent of damage (e.g. fatigue cracks, corrosion, impact damage, etc.)
4. Load levels and fatigue spectrum experienced by the structure
5. Material thickness to be repaired
6. Skill of the available labor
7. Availability of repair materials
8. Repair facility

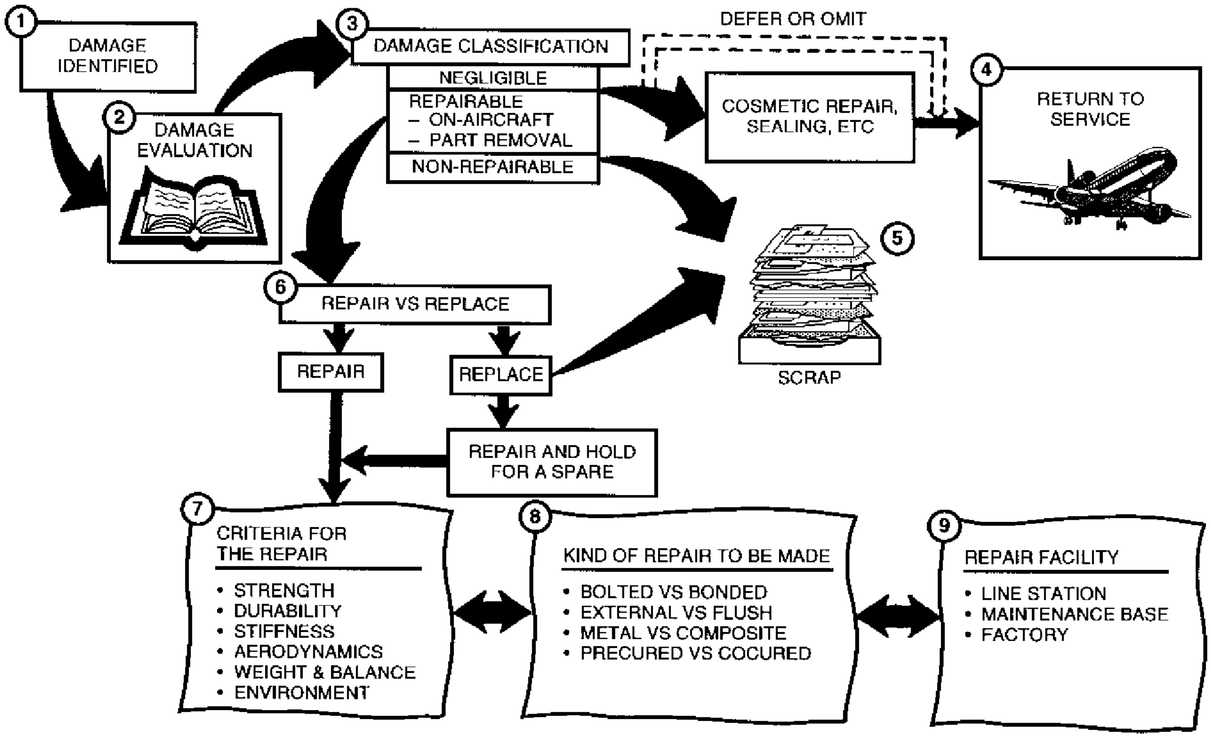


Figure 7. Damage Evaluation and Selection of Repair Methods

4. REPAIR OPTIONS

4.1 Repair of Composite Materials

Repairs of composite materials are similar to those for metallic materials if mechanically fastened repairs are to be used. However, the repairs of composite materials are different from those of metals if the repairs are to be bonded. The damage must be evaluated and classified. If the damage is repairable, a decision has to be made whether to repair or replace a part. If the structure is to be repaired, additional decisions have to be made regarding maintenance level, where work will be done, kind of repair materials, and repair configuration. The first step in the repair of composite materials is to remove the damage area including the delaminated area in the impacted region. The next step is to clean the surface to be repaired and apply a bolted or bonded patch. These repair concepts are discussed in the following paragraphs.

BOLTED REPAIRS

Bolted repairs for composite structures are similar to those for metallic structures. The major differences between the repairs for composites and metals are:

a) Different tools are used for drilling fastener hole in composites.
b) Special care is needed in drilling holes in composites to prevent splintering on the exit side of the hole. A back support is desirable.
c) Matrix in composite is brittle compared to metal, hence the fasteners that expand to fill the hole (e.g. driven rivets) are not suitable for composites.
d) Sharing of loads in different fasteners in composites is not uniform because composite materials do not yield as metals where the load distribution tends to be more uniform.

Three commonly used bolted repair concepts are shown in Figure 8 and are discussed here.

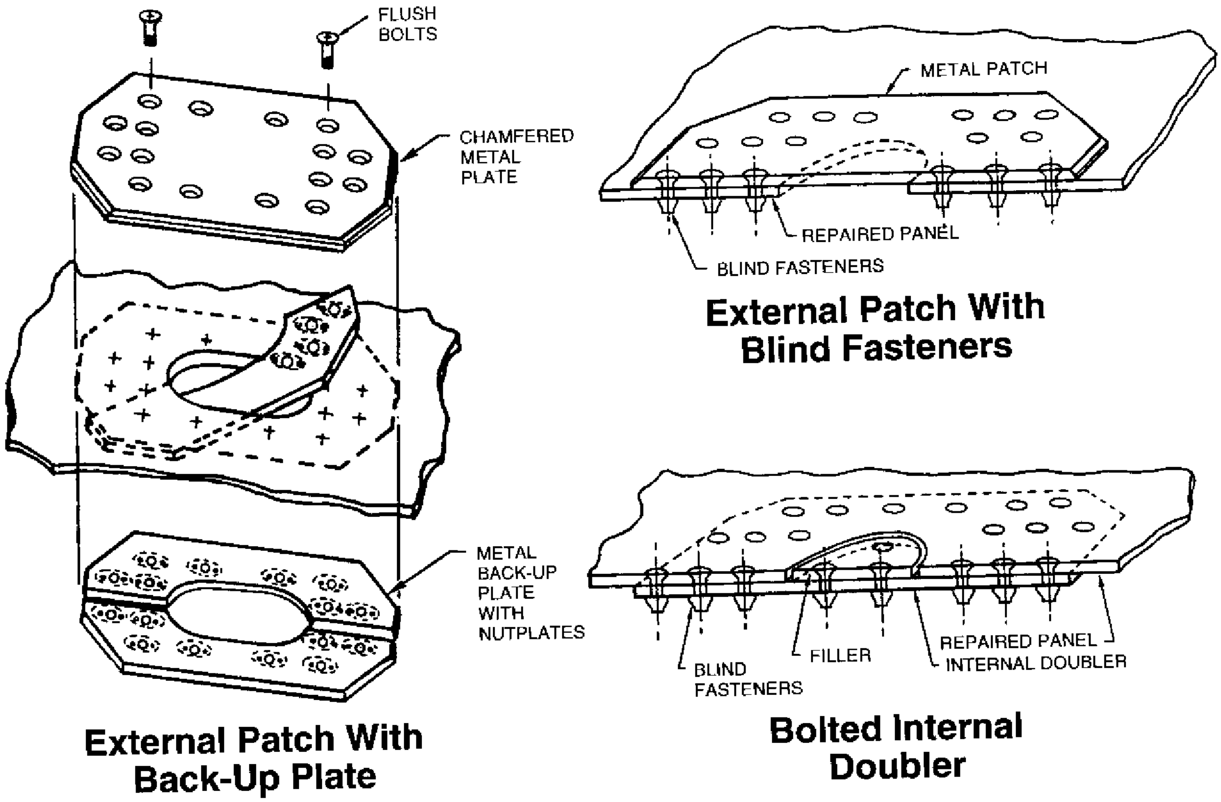


Figure 8. Bolted Repair Concepts

External Patch with Backup Plate- This concept uses an external chamfered metal patch bolted to the panel being repaired as shown in Figure 8. The bolts thread into nut plates mounted on metal backup plates that are on the side of the repaired panel. The backup plate can be split into two or more pieces and slipped through the opening as shown in the figure.

External Patch with Blind Fasteners-This concept is similar to the previous one, except that the backup plates are not used as shown in Figure 8. Blind fasteners are not as strong as bolts and nutplates, but if acceptable strength can be restored, this concept is easier to use.

Bolted Internal Doubler-This concept has been used as a standard repair for metal structures. Access to the backside is required to install the doubler as shown in Figure 8. The doubler cannot be installed through the hole as a separate piece because the doubler has to be continuous to carry loads in all directions. Filler is used to provide a flush outer surface, and is not designed to carry loads.

BONDED REPAIR CONCEPTS

Bonded repair concepts can restore greater strength to a damaged composite structure as compared to bolted repairs. External repair patches are suitable for thin skins, however, for thick skins the eccentricity of the external patch reduces its strength. Flush patches are preferred for thick structures, heavily loaded structures, or where aerodynamic smoothness is required. Commonly used repair concepts are step-lap and scarf repairs.

Step-Lap Repair- This repair concept is shown in Figure 9. The steps allow the load to be transferred between specific plies of the patch and parent material. This advantage tends to increase the strength of the joint; however, it is offset by the peaks that exist in the adhesive shear stress at the end of each step.

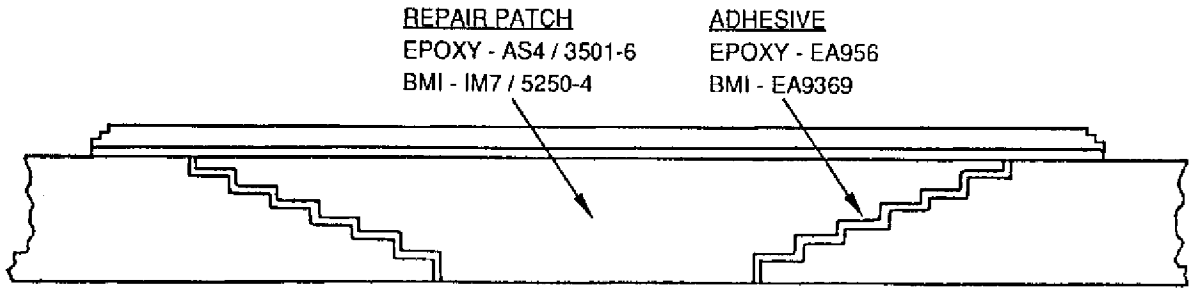


Figure 9. Step-Lap Repair

Scarf Repair- This repair concept is shown in Figure 10. The patch material is within the thickness to be repaired, with additional external plies added for strength. This configuration can restore more strength than an external patch as it avoids the eccentricity of the load path and provides smooth load transfer through gradually sloping scarf joint. A properly designed scarf joint can usually develop the full strength of an undamaged panel. The patch material is usually cured in place, and therefore must be supported during cure. While the patch material can be cured and then later bonded in place, it is generally difficult to get a good fit between the precured patch and the machined opening. In practice, well-made step-lap and scarf joints have approximately the same strength. A disadvantage of step-lap joints is the difficulty in machining the step to the depth of the exact ply that is desired on the surface of the step.

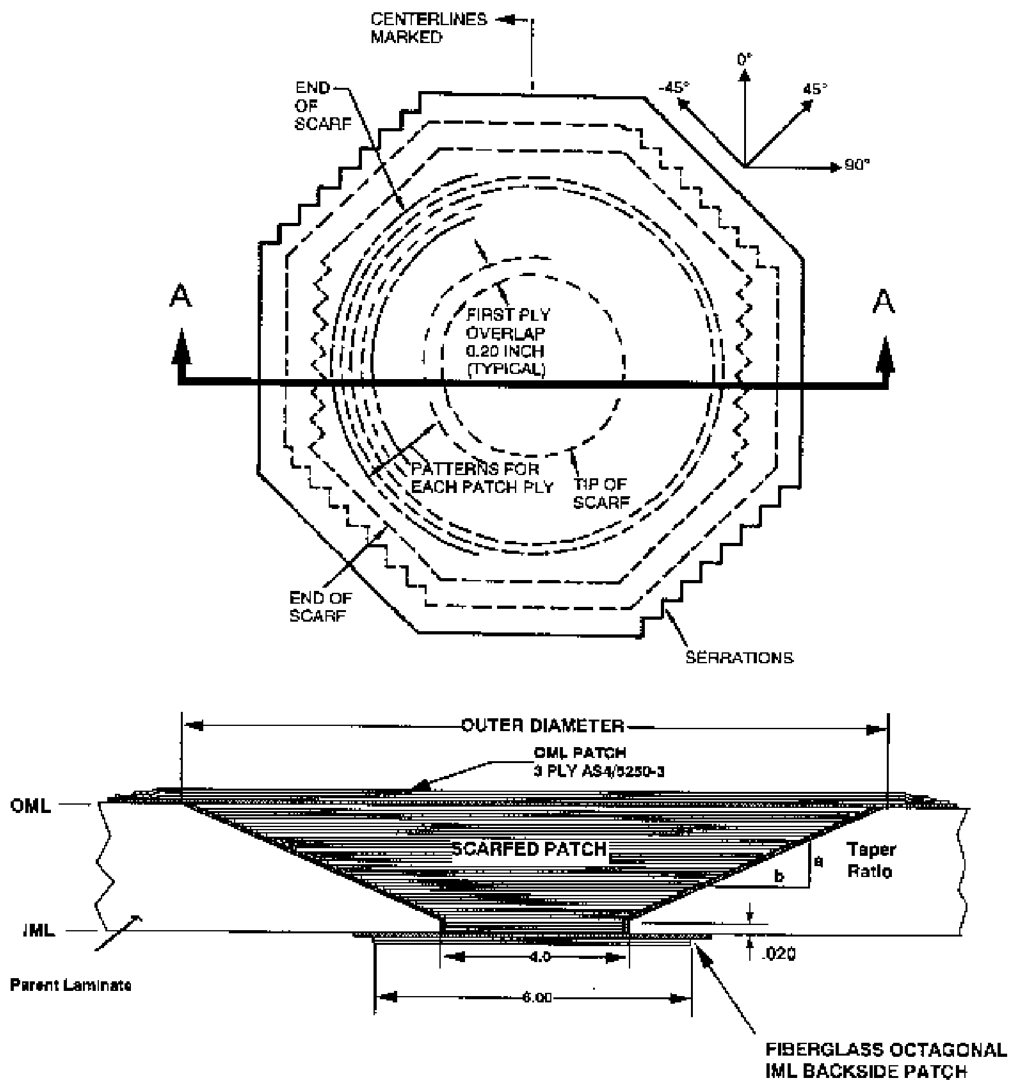


Figure 10. Scarfed Repair

4.2 Repair of Sandwich Structures

A typical in-service damage to a sandwich structure with composite face sheets is shown in Figure 11. The damage to composite face sheets is visible damage with surface indentation. Delaminations are seen in the composite face sheets as well as disbonding between the face sheets and honeycomb core. In addition, core buckling is seen.

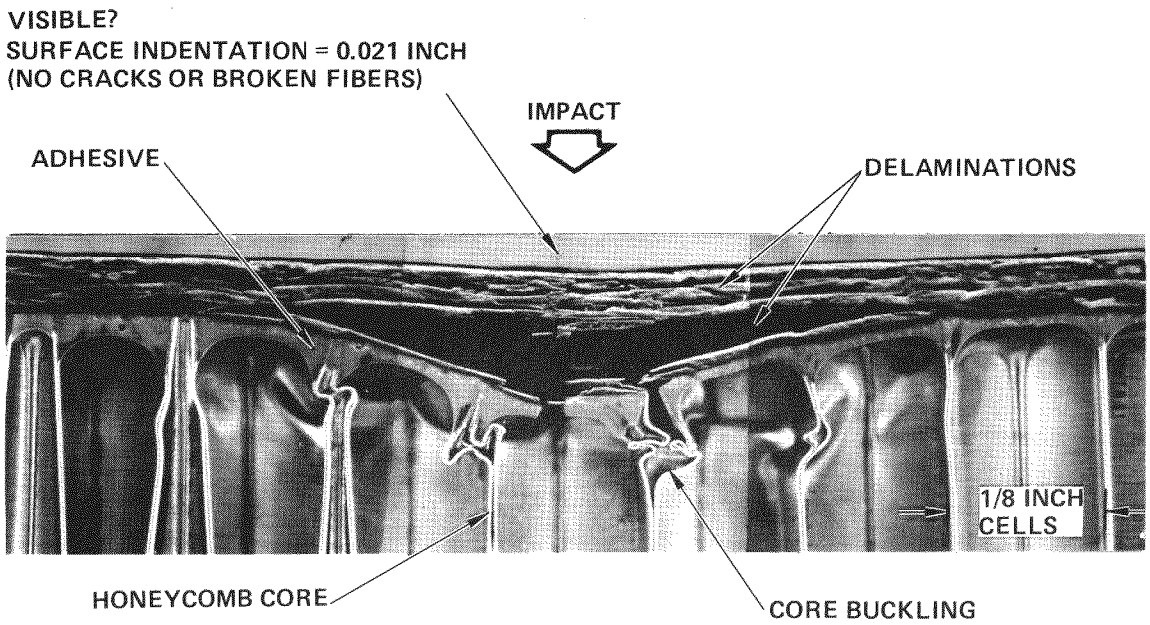


Figure 11. Typical Impact Damage in Sandwich Structure with Composite Face Sheets

The repair of a sandwich structure will depend on the extent of the core damage. Full depth and partial through the depth repair concepts are shown in Figure 12. The core damage has to be machined out and a plug prepared before performing the repairs. Various steps involved in the repair are illustrated in the figure.

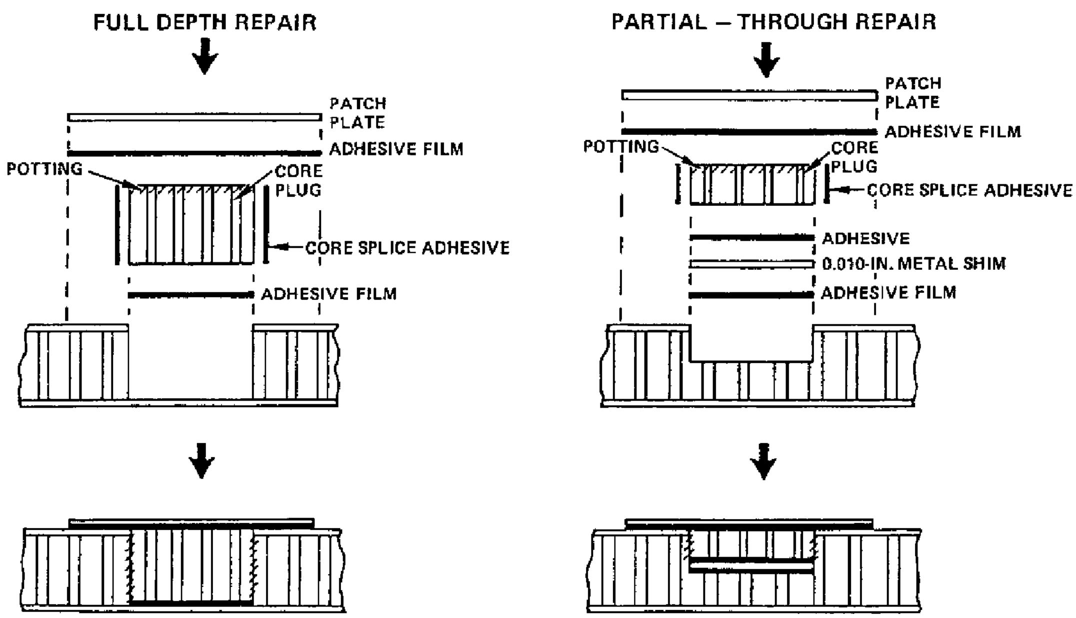


Figure 12. Repair Concepts for Sandwich Structure with Composite Face Sheets

4.3 Repair of Metallic Structures

4.3.1 MECHANICALLY FASTENED REPAIRS OF METALLIC STRUCTURES

Repair concepts for metallic structures are well established. The bolted repair concepts, discussed earlier for composites are applicable to metallic repairs. Standard repairs are generally given in repair manuals. However, in many cases in-service inspections show damages that are not covered by standard repair manuals and special repairs have to be designed. For such cases detailed static and damage tolerance analyses have to be carried out. An example of cracked frame in a transport aircraft (Figure 13) is shown in Figure 14. The flange and the web of the frame are cracked as shown in Figure 15a. Standard repair manuals generally do not cover a repair for the damage shown in Figure 14. The cross-sections of the flange and web repairs are shown in Figure 15b. The details of the frame repair are shown in Figure 16.

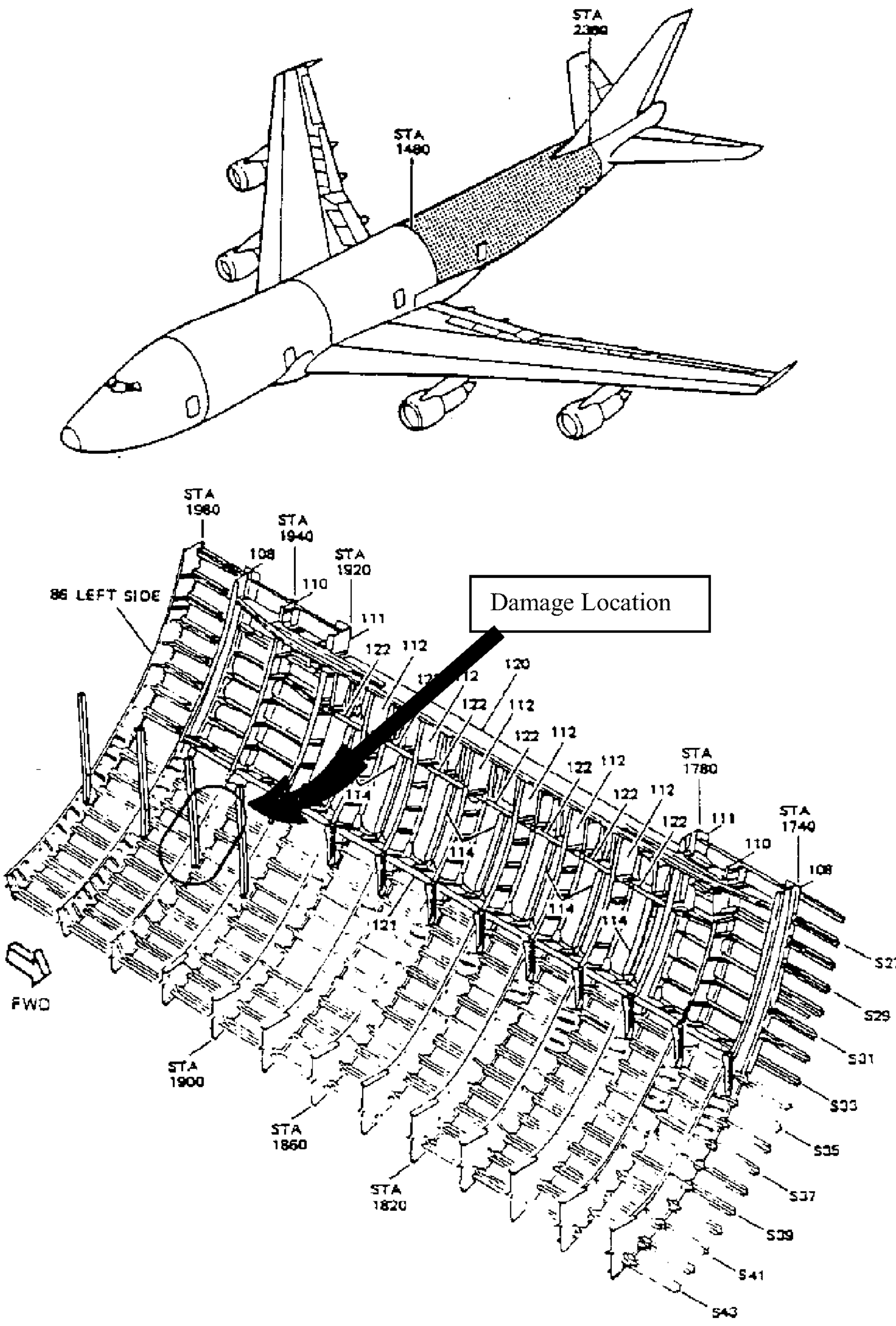


Figure 13. Cracking Location in Transport Aircraft Fuselage

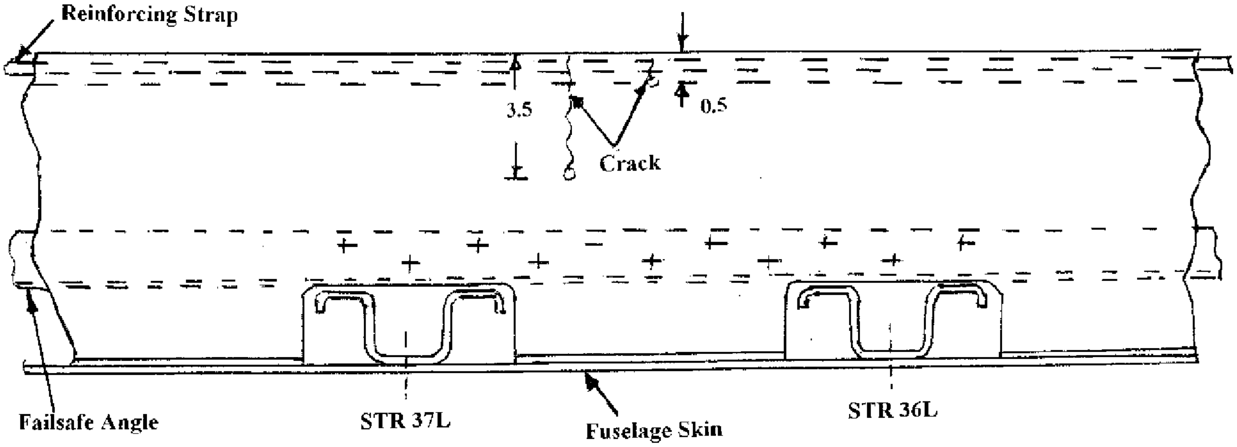
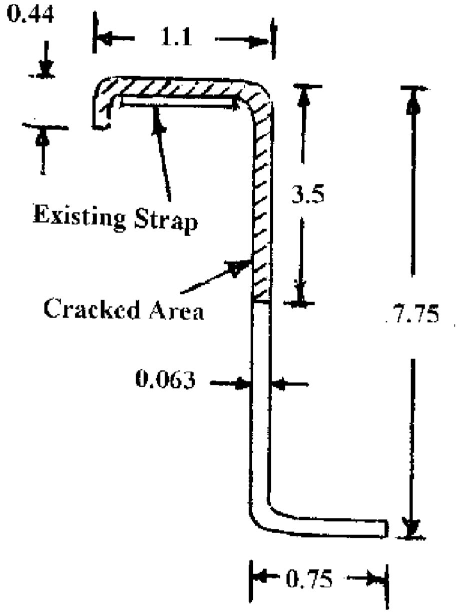


Figure 14. Cracked Frame



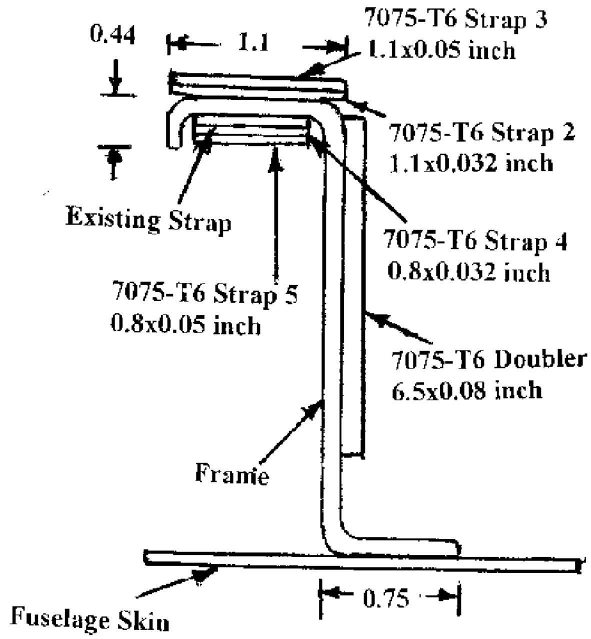


Figure 15a. Cross-section of Cracked Frame Figure 15b. Cross-section Showing Flange and Web Repair

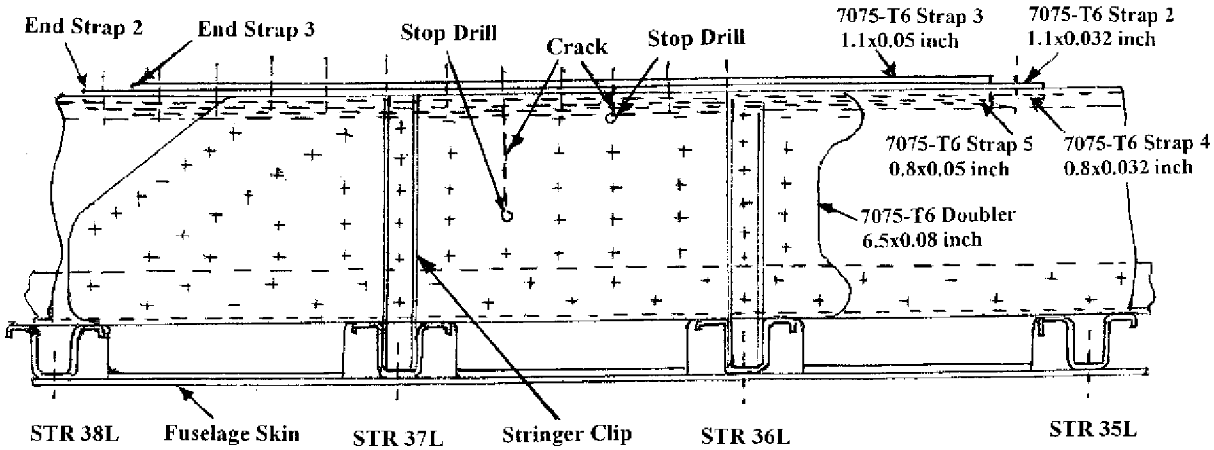


Figure 16. Details of Frame Repair

4.3.2 BONDED REPAIRS OF METALLIC STRUCTURES

The conventional mechanically fastened repair concept has disadvantages primarily due to the drilling of holes for additional fasteners that affect the structural integrity of the structure. In many cases the parts have to be scrapped due to the repaired structure not meeting the fail safety requirements. In most cases if the thinning due to corrosion is more than 10% of part thickness the parts are replaced. The development of bonded composite repair concept has provided excellent opportunities to design more efficient repairs (References 4, 6-13) and in many cases has made it possible to repair damaged structures which could not be repaired with the conventional mechanical fastening and were scrapped. Composite patch repairs also result in reduced inspection requirements compared to mechanically fastened repairs. In fact, in many cases the composite patch repairs can be designed such that the cracks in metallic structures underneath the repairs will not grow thereby eliminating inspection requirements, except those imposed by Integrated Logistics Supports (ILS) plan.

In bonded composite repair concept a composite patch is bonded to the damaged metallic part instead of a conventional mechanically fastened patch. Bonded composite repair has many advantages over conventional mechanically fastened repair, namely: 1) More efficient load transfer from a cracked part to the composite patch due to the load transfer through the entire bonded area instead of discrete points as in the case of mechanically fastened repairs, 2) No additional stress concentrations and crack initiation sites due to drilling of holes as in the case of mechanically fastened repairs, 3) High durability under cyclic loading, 4) High directional stiffness in loading direction resulting in thinner patches, and 5) Curved surfaces and complex geometries easily repairable by curing patches in place or prestaging patches. The cross-section of a typical 16-ply graphite/epoxy patch bonded to an aluminum sheet is shown in Figure 17.

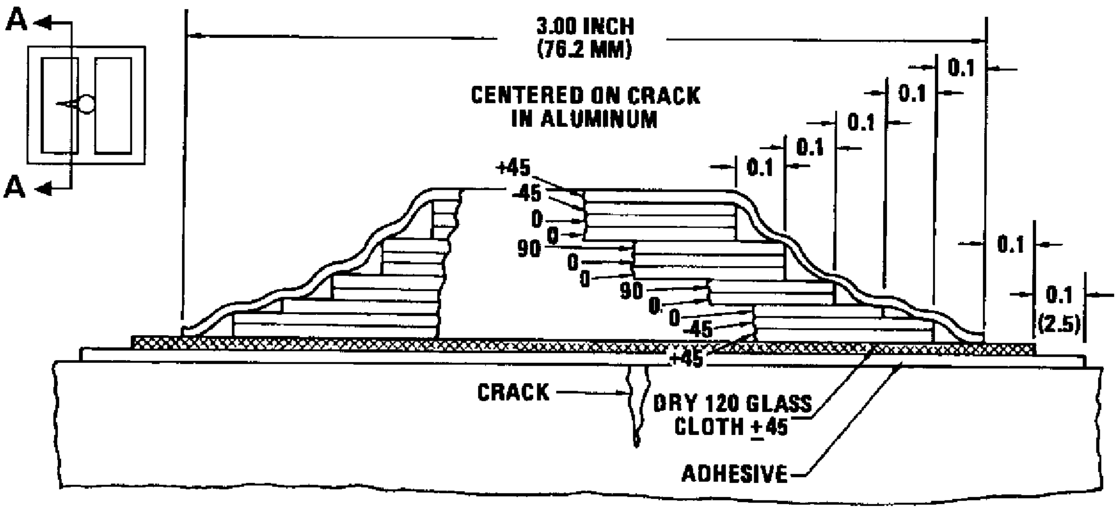


Figure 17. Cross-section of a Typical Composite Repair Patch

The critical parameters for this type of repair are 1) Surface preparation, 2) Adhesive material selection, 3) Composite repair material selection, and 4) Bonding operation.

Surface Preparation

Proper surface preparation is one of the most important considerations in bonded structures. The surface preparation process consists of paint removal, anodizing and priming. Liquid chemical paint strippers are not recommended, as they may become entrapped in cracked areas and faying surfaces of adjoining structures, thereby causing a corrosion problem. Aluminum oxide abrasive cloth has been found to be suitable for small repair areas.

Both silane and phosphoric acid non-tank anodize (PANTA) have been found to be suitable. The silane process has the advantage of being non-acid process. However, from the point of view of long term durability of repairs, the PANTA process may be desirable, as sufficient test data is available on this process.

Primer is applied to the aluminum surface after anodizing with PANTA to prevent contamination and improve long-term durability. BR-127 primer has been found to be suitable for FM-73 adhesive.

Adhesive Material Selection

Room temperature cure adhesives are not considered suitable due to service temperature requirements of 180F (82C) in the majority of aircraft repair applications. Also, room temperature cure adhesives are paste adhesives and generally do not result in uniform bond line thickness in the repair. Thus, affecting the load transfer to composite patch. Hence, high temperature film adhesives are preferred. Also, long term durability of room temperature adhesives is not well characterized. A 350F (177C) cure film adhesive is not considered desirable, as the curing at such a high temperature is likely to cause undesirable high thermal stresses. Also, an aluminum structure exposed to a 350F (177C) temperature will undergo degradation in mechanical properties. A 250F (121C) cure adhesive system is considered suitable for the composite patch repair of aluminum structure. Ductile adhesives such as FM-73 are preferred over brittle adhesives such as FM-400 due to the tendency of the brittle adhesives to disbond around the damage area, thereby reducing the load transfer to the repair patch.

Composite Repair Material Selection

Both boron/epoxy and graphite/epoxy composites are suitable for the repairs. The choice between boron or graphite fibers should be based on availability, handling, processing and the thickness of the material to be repaired. Boron has higher modulus than graphite and would result in thin repair patches. Thin patches are more efficient in taking load from damaged parts as compared to thick patches. For repairing relatively thick parts, boron may be preferred over graphite. It is considered desirable to use highly orthotropic patches, having high stiffness in the direction normal to the crack, but with some fibers in directions at 45 and 90 degrees to the primary direction to prevent matrix cracking under biaxial loading and inplane shear loads which exist for typical applications. This patch configuration can be best obtained with unidirectional tape. Woven material has greater formability and could also be used, although it would not make a very efficient patch.

The composite patches may be precured, prestaged or cured in place. For locations where vacuum bagging represents a problem, a precured patch may be prepared in an autoclave and then secondary bonded to the repair area. For relatively minor contours, a prestaged patch may be used. For curved surfaces the patch may be cured in place during the bonding operation.

Bonding Operation

Bonding of repair patches requires a proper temperature control within +10F and -5F in the repair area. Thermal blankets are available to provide temperature in excess of 1000F (538C). A proper temperature control within tolerances is necessary for bondline to achieve desirable strength. A large aircraft structure compared to a small repair area may act as a heat sink and jeopardize maintaining desired temperature control for the required duration. Proper heat blankets for surrounding areas may be required for such cases.

Crack Growth Life Enhancement with Bonded Composite Repairs

The crack growth data obtained from a repaired center-crack panel (7075-T6 aluminum, 0.063-inch (1.6-mm) thickness) are shown in Figure 18. It is seen that starting with the same initial crack length, the panel without a repair patch fails after about 870 missions (0.92 lifetime) at a crack length of 1.36-inch (34.6-mm). The panel with the repair patch did not fail even after 2350 missions (2.5 life times) at a crack length of 1.93 inches (49 mm). Thus, a considerable extension in life was obtained with the composite repair patch.

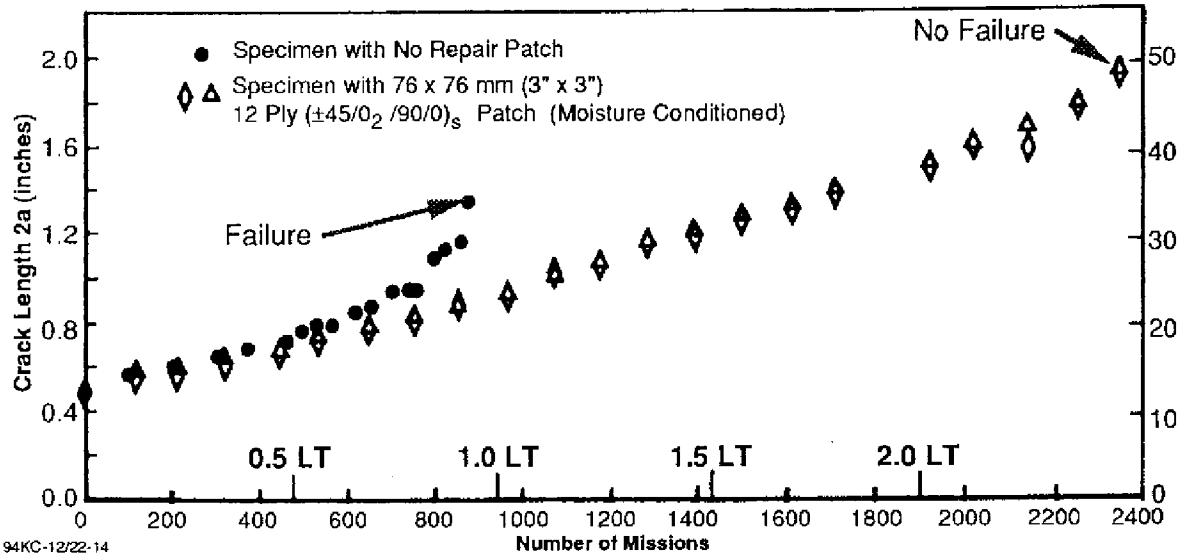


Figure 18. Comparison of Crack Growth in Specimen With and Without Repair Patch

Comparison of Analytical and Experimental Results

The crack growth behavior of the cracked panel with a composite patch was predicted using analytical stress intensity factors (Ref. 14-15) for the patched structure and the crack growth data, obtained on an unpatched center crack specimen. Comparison of observed and predicted fatigue crack growth behavior in a 7075-T6 aluminum 0.063 inch (1.6 mm) thickness repaired with a 3 inch (76 mm) square 12 ply graphite/epoxy patch, moisture conditioned to one percent moisture, is shown in Figure 19. It is seen that the correlation between predicted and observed crack growth is excellent. The specimen did not fail even after two life times of spectrum loading.

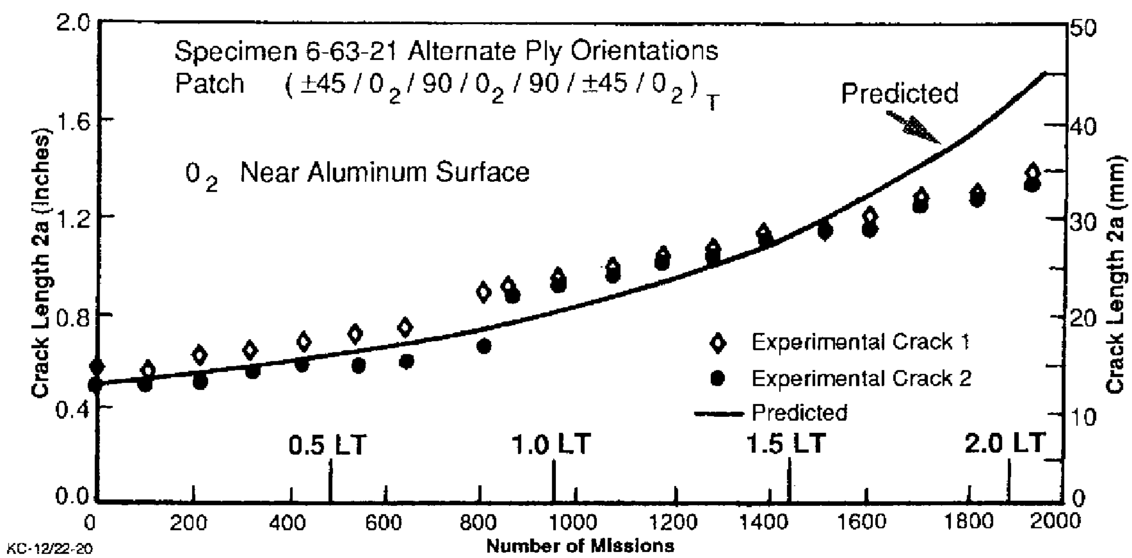


Figure 19. Comparison of Observed and Predicted Crack Growth

Repair Design for No Damage Growth

It is possible to design composite repair patches so that the damage in the repaired structure will not grow. Of course, the feasibility of such a design depends on the stress level, the type of material to be repaired, material thickness, the crack length to be repaired, and spectrum. In the majority of transport aircraft where design stress levels are relatively low, it is possible to design repairs such that the damage does not grow. This is particularly true for fuselage structures where material is predominantly 2024-T3 aluminum and gauge thicknesses are small. Crack growth behavior in 2024-T3 material 0.032-inch (0.8-mm) thick specimen, repaired with 12-ply Gr/Ep patch is shown in Figure 20. No crack growth in two lifetimes of spectrum loading is seen. Thus, the repairs can be designed for no damage growth and there by eliminating inspection requirements.

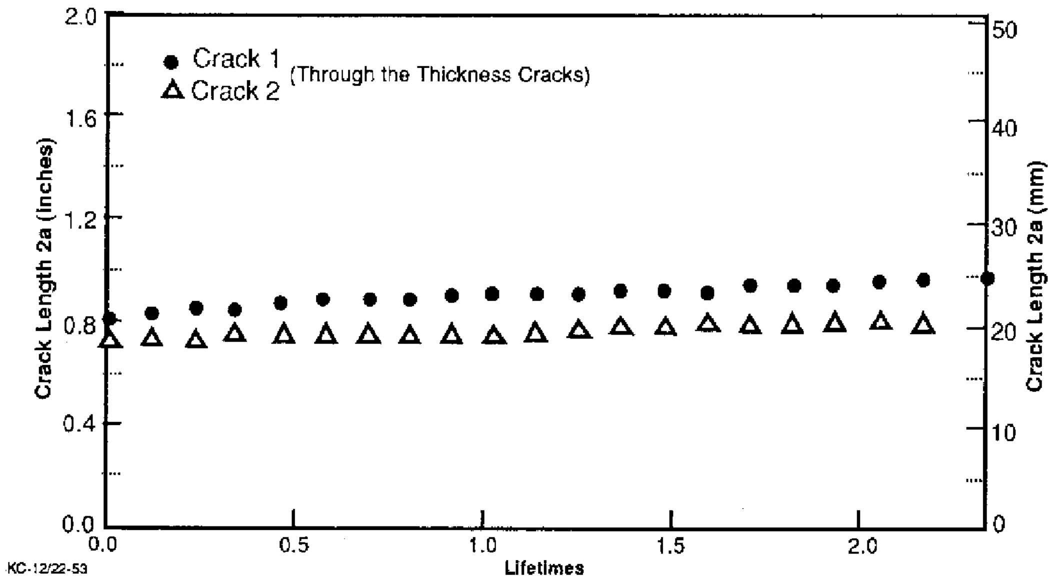


Figure 20. Crack Growth in 2024-T3 Aluminum, 0.032 inch (0.8 mm) Thick With 12-Ply Gr/Ep Patch

4.3.3 IN-SERVICE APPLICATIONS OF COMPOSITE PATCH REPAIRS

Applications of composite patch repair to in-service aircraft are found in T-38 lower wing skin (References 16-19), C-141 weep holes (Reference 20) and F-16 fuel access hole (Reference 21). T-38 lower wing skin has developed in-service cracking problems at "D" panel attachment holes and at machined pockets between 39% and 44 % spars and 33% and 39% spars as shown in Figure 21. Composite patch repair concepts were developed for these locations.

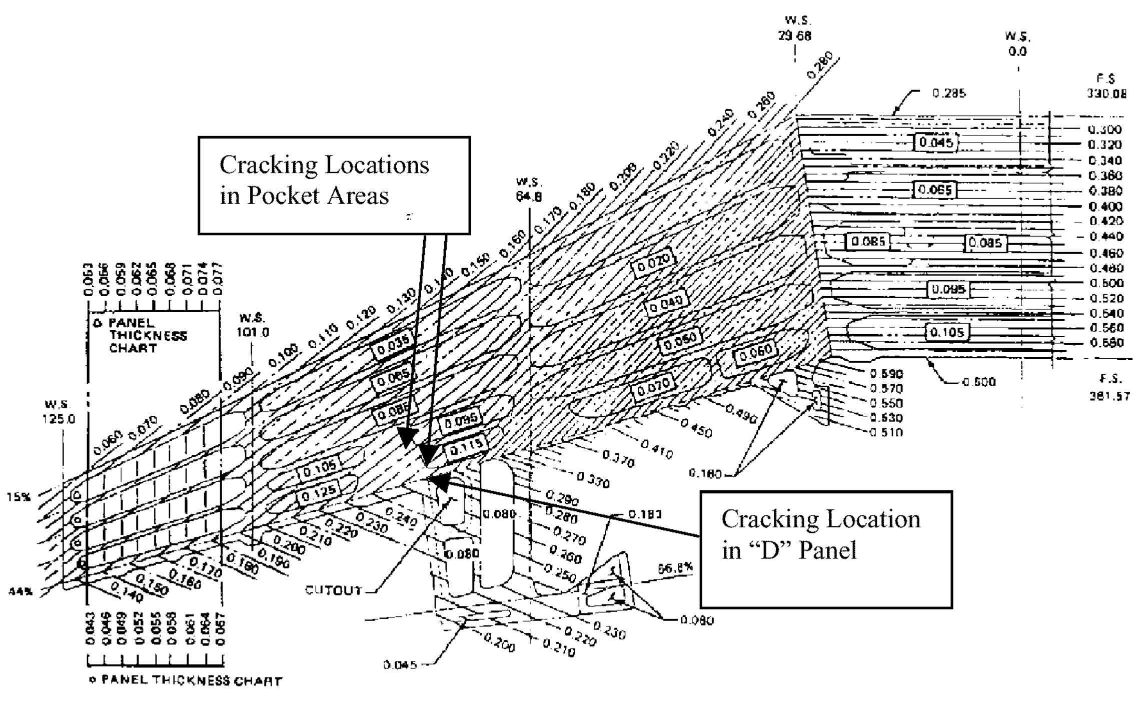


Figure 21. Cracking Location in T-38 Lower Wing Skin

Conventional mechanically fastened repair concepts at the location of "D" panel are not possible due to the limited space available for drilling the fastener holes. Bonding of an aluminum doubler will provide only limited doubler stiffness and will not result in an efficient repair. A bonded boron repair is ideal for this location. An external boron patch could not be applied as the door has to fit in the area and has to be flush with the outer mold line. Hence, an internal repair patch was designed as shown in Figure 22. A pre-cured boron repair patch was secondary bonded through the 'D' panel door.

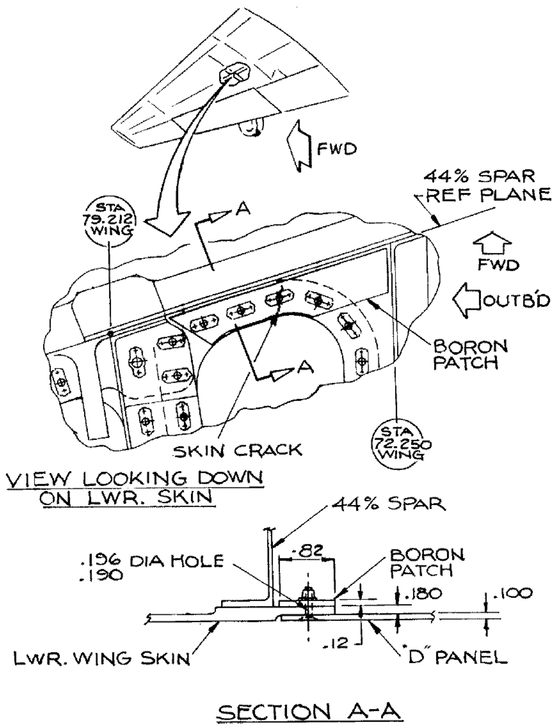


Figure 22. T-38 Lower Wing Skin Composite Patch Repair

Lower wing skin pockets in T-38 aircraft between the 39% and 44% spars and 33% and 39% spars at Wing Station (WS) 78 have shown a propensity for crack initiation and propagation during service. The cracks have initiated at the pocket radius in the inner moldline of the wing skin. This cracking has been occurring primarily under Lead-in-Fighter (LIF) spectrum loading. These areas are ideal for composite reinforcement to reduce stress levels and enhance fatigue life. As there is no access for bonding reinforcement on the inner moldline, a one sided reinforcement bonded onto the outer moldline of the wing skin was selected. Due to the complexity of the structure in the area, it was considered necessary to verify the reinforcement design by structural testing. The test program was devised in two parts. In the first part of the test program, testing was performed on specimens that simulate the configuration and load environment in the pocket areas of the wing. The results of this study are reported in Reference 18. The second part of the test program involved bonding of the reinforcement to a T-38 wing (Figure 23) subjected to durability testing at Wright Patterson Air Force Base (WPAFB), Ohio, as a part of Air Force Contract F33615-90-C-3201, entitled "Advanced Technology Redesign of Highly Loaded Structures (ATROHS)". The wing with composite reinforcement has undergone 3,500 hours of testing under LIF spectrum loading (Reference 17).

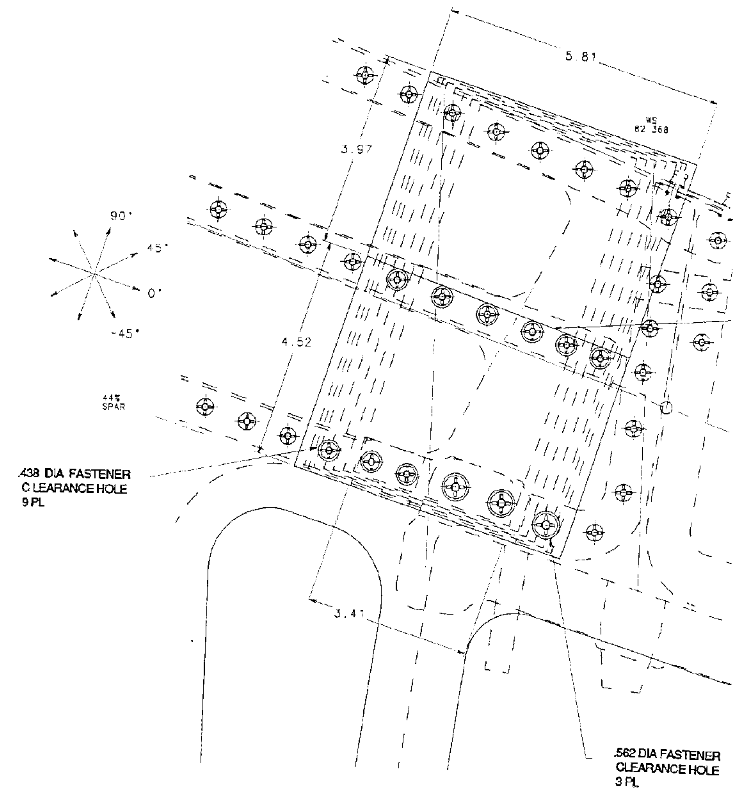


Figure 23. Composite Reinforcement in Lower Wing Skin Pocket Areas

Vacuum-bagged composite reinforcement assembly on T-38 test wing is shown in Figure 24 and bonded reinforcement assembly is shown in Figure 25.

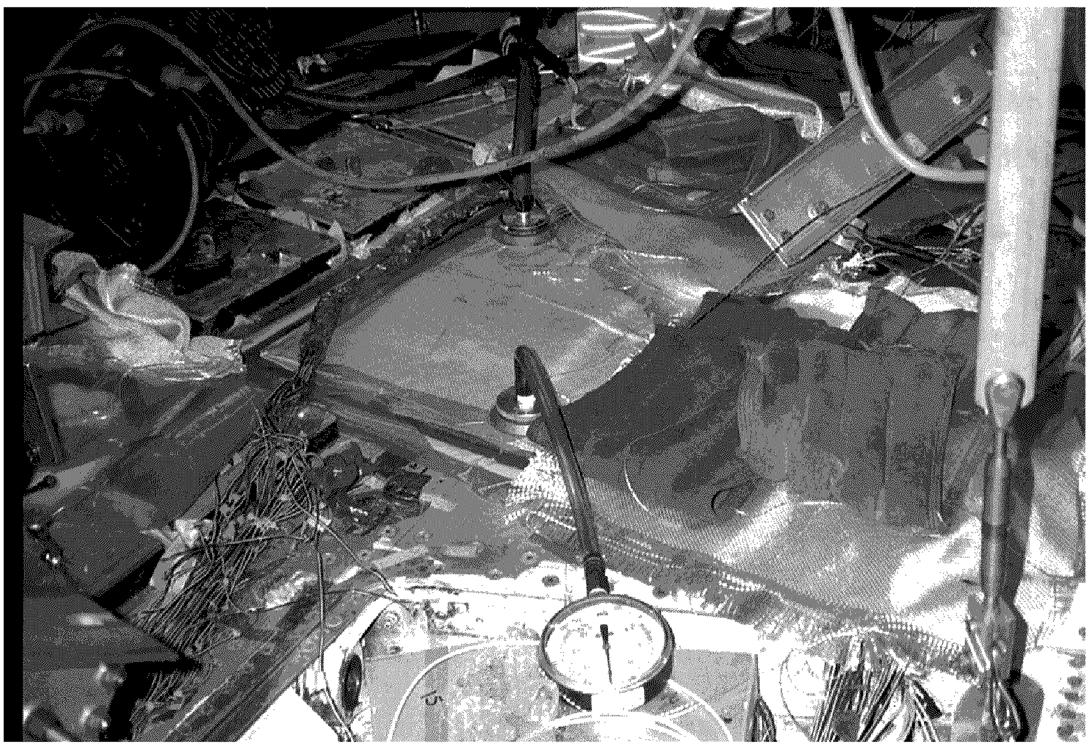


Figure 24. Vacuum Bagged Reinforcement Assembly on T-38 Test Wing

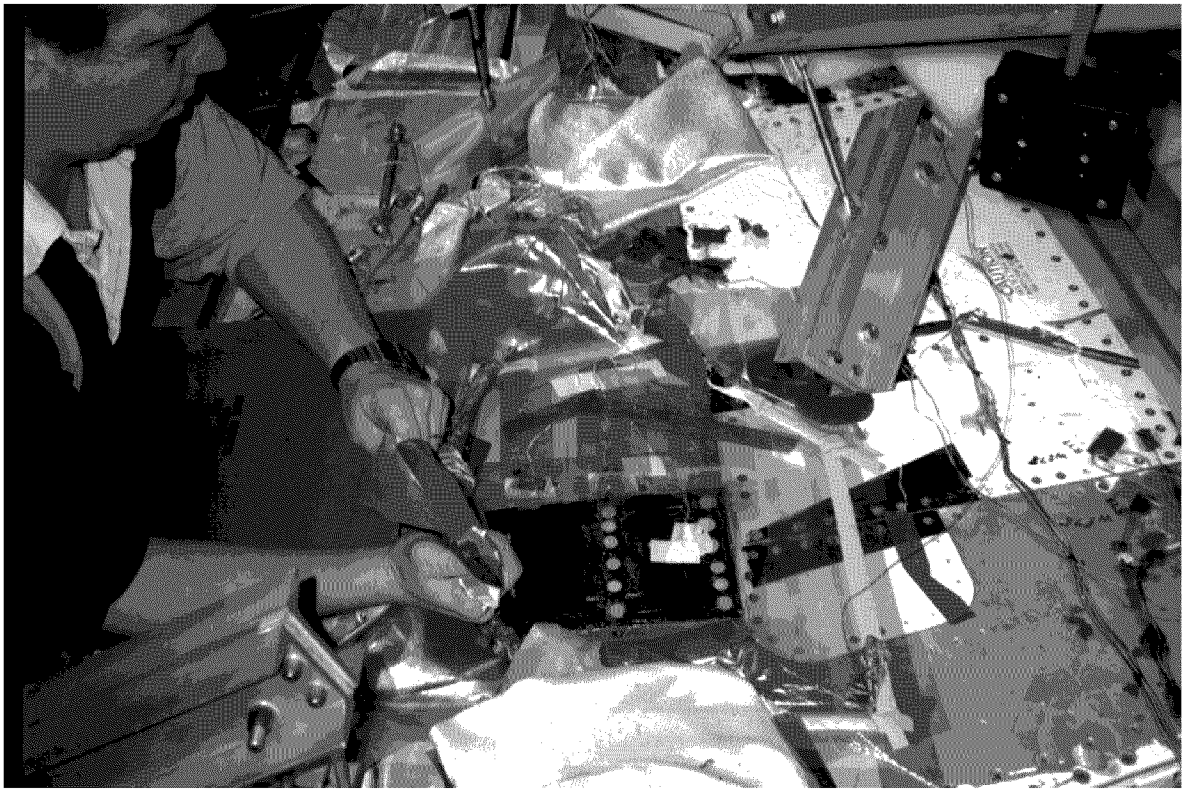


Figure 25. Bonded Reinforcement Assembly

Composite patch repair application to C-141 lower wing skin at weep holes is shown in Figure 26. Figure 27 shows composite reinforcement application to lower wing skin splice area.

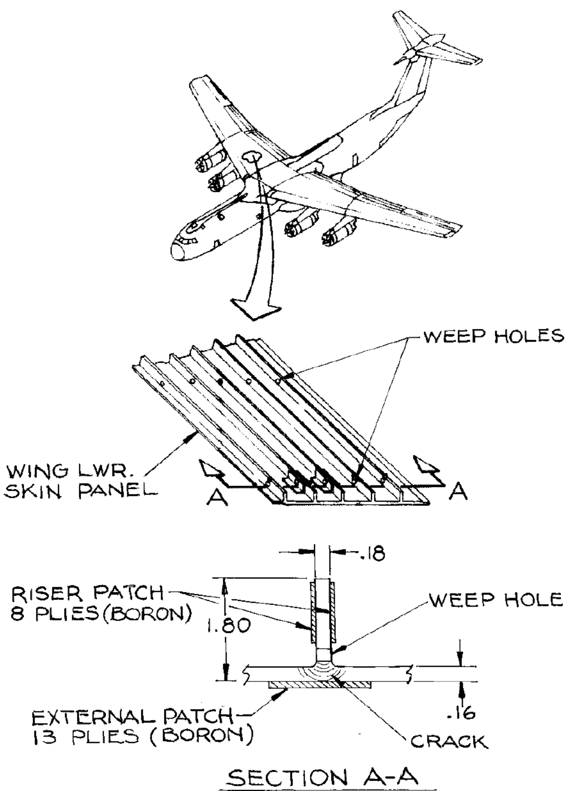


Figure 26. C-141 Composite Patch Repair at Weep Holes

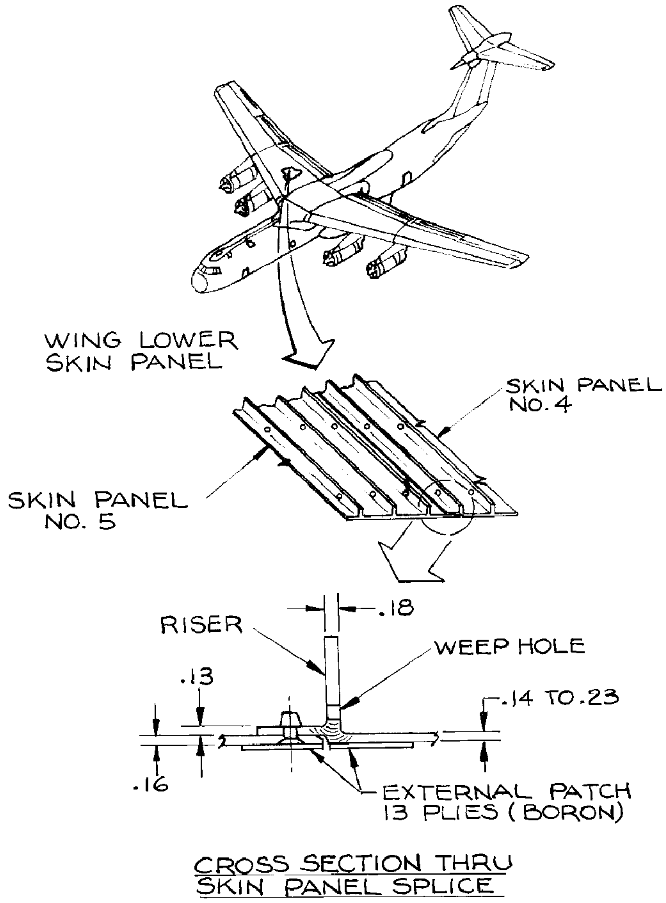


Figure 27. Composite Patch Repair at C-141 Lower Wing Skin Splice

5. SOFTWARE FOR REPAIR DESIGN AND ANALYSES

A number of software programs have been developed for designing repairs for aircraft structures. Some of these programs are briefly described here.

1. RAPID- This program has been developed under FAA and US Air Force sponsorship and is primarily for mechanically fastened repairs of transport aircraft. The program has capability to perform analysis under spectrum loading.
2. RAPIDC- This program has been developed under FAA sponsorship and is primarily for mechanically fastened repairs of commuter aircraft.
3. AFGROW- This is US Air Force developed code for durability and damage tolerance analyses of aircraft structures under spectrum loading. This code has capability to design composite patch repairs.
4. CalcuRep- This code has been developed by Dr. Rob Fredell during his stay at US Air Force Academy in Colorado. This code is for designing bonded repairs, using GLARE, for fuselage type of structures.
5. FRANC2D- This is a finite element code and can be used for damage tolerance analysis and composite patch repair design under constant amplitude loading.
6. COMPACT3D- This is a finite element code for designing composite patch repairs under constant amplitude loading.
7. NASGRO- This program has been developed by NASA Johnson Space Center and is available in public domain. The program is primarily for damage tolerance analyses.

6.0 CONCLUDING REMARKS

The life enhancement technologies have provided excellent opportunities to fulfill aging aircraft needs such as:
1) Reduced life cycle costs
2) Reduced/eliminated repairs
3) Reduced/eliminated inspections
4) Simplified maintenance

5) Reduced support requirements
6) Fulfilled severe usage requirements
7) Extended airframe life
8) Improved payload

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